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AUTHORITY

USNATTS ltr, 6 Jan 1966

U. S. NAVAL AIR TURBINE TEST STATION

TRENTON, NEW JERSEY

Aeronautical Turbine Laboratory

NATTS-ATL-92 June 1964

AN INVESTIGATION OF THE SUPERSONIC RESEARCH CAPABILITIES OF THE NATTS ALTITUDE TEST FACILITIES

Phase I: Preliminary Engineering Considerations

Foundational Research Project FR-11

Prepared by:

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INTRODUCTION

The consistent advancement of the "state of the art" in the areas of air breathing propulsion systems and aerodynamics requires continuous evaluation and improvement of the test facilities used in these fields. The purpose of this study was to evaluate the capacity of the Naval Air Turbine Test Station's altitude test facility in terms of supersonic aerodynamic research and free jet air breathing engine testing. Appendix A provides details of the initial project proposal. The evaluation reported herein was made to determine:

(1) The extent of supersonic operation available.

(2) The effect of various tunnel diffuser configurations.

- (a) Straight pipe.
- (b) Constant geometry, convergent-divergent.
- (c) Variable geometry, convergent-divergent.

(3) The magnitude of the Reynolds number per inch parameter.

(4) The approximate air breathing engine size that may be accommodated.

(5) A configuration and the values for certain parameters in a design point tunnel.

The direct connected technique of engine testing is presently used at NATTS. It is quite suitable for subsonic flight conditions, but requires some assumption regarding total pressure recovery for supersonic flight conditions. In an effort to avoid the errors involved, this evaluation considered only the free jet type of installation. As Mach number increases within the supersonic regime, it becomes more and more difficult to study the engine and inlet separately.

Some unpublished studies of free jet testing for subsonic flight conditions have been made. Results indicated that the excess air required for operation of present day engines was beyond facility limitations. For a subsonic configuration, downstream blockage effects due to the engine were transmitted upstream where they disturb the flow field. Large amounts of excess air were required to make these effects negligible. Since no effects are

transmitted upstream in a supersonic flow, a smaller portion of excess air would be required. The supersonic free jet installation requires only that no pressure waves or their reflections enter the engine inlet. With due consideration of these facts, the study of the supersonic capabilities of the NATTS facility appeared to be a logical next step.

Once a supersonic capability has been demonstrated, the uses for it will be many and varied:

(1) Free jet test and development for air breathing engines.

(2) Basic and applied aerodynamic research.

(3) Engine inlet and nacelle studies.

(4) Individual studies of inlet diffusers and exhaust nozzles.

(5) Combustion in supersonic flows.

(6) Behavior of instrumentation in supersonic flows.

The capacity of the 3E test cell in-line heater has been incorporated into this study since it will be available in the near future. It can provide true temperature or standard altitude temperature simulation over a much greater Mach number range than the "C" rig heater available in the 1E and 2E cells.

SUMMARY OF RESULTS

1. The NATTS altitude test facility was capable of supersonic operation at Mach numbers up to 5.0. At a stagnation pressure and temperature of 55 in. HgA and 640°R, the test section diameter was on the order of five feet for the maximum available airflow. The use of standard altitude temperature increased the test section diameter to approximately six feet for the same airflows.

2. The use of various tunnel configurations had no effect on the maximum airflow for a given size tunnel (maximum airflow was a function of the starting process) but did affect the percent plant capacity being used. At a Mach number of 5.0, a variable area convergent-divergent diffuser required only 20 percent of the available exhauster capacity. At the same Mach number, a constant geometry convergent-divergent diffuser and a straight pipe diffuser required 70 and 100 percent respectively of the available exhauster capacity.

3. Reynolds number values per inch below 750,000 at a Mach number of 5.0 were available at stagnation temperatures above 1060°R for a stagnation pressure of 55 in. HgA. If the stagnation pressure was reduced to 35 in. HgA, these Reynolds number values per inch were possible at stagnation temperatures of 640°R and above at a Mach number of 5.0.

4. At a stagnation pressure and temperature of 55 in. HgA and 640°R respectively, a closed type free jet installation could accommodate a ramjet with a five square feet maximum cross sectional area at a Mach number of 4.0. Use of an open type of free jet (i. e., test section area greater than inlet nozzle exit area) increased the maximum engine cross sectional area to approximately nine square feet for the same Mach number. For the same stagnation pressure but using standard altitude temperature variation, these areas were increased to seven and nineteen square feet respectively.

5. Using the standard altitude temperature variation and a stagnation pressure of 55 in. HgA, a turbojet engine whose sea level static airflow was less than or comparable to a J52 or J79 engine could be operated at Mach numbers up to 2.5 if a suitable (i.e., supersonic) inlet and an open type of free jet configuration were used.

6. The use of a fixed inlet stagnation pressure restricted the available test conditions to a given altitude and Mach number variation.

LIST OF SYMBOLS

Symbol	Definition	Units
A	Area	ft^2
M	Mach number	---
K	Constant	$.532°R^{1/2}/lb$
P_o	Stagnation pressure	in. HgA; lb/ft^2
p	Static pressure	in. HgA; lb/ft^2
T_o	Total temperature	°R, °F
T	Static temperature	°R, °F
W_a	Airflow	lb/sec
α	Inlet area/maximum engine area	---
β	Plug area/maximum engine area	---
γ	Inlet diffuser area ratio	---
η	Adiabatic diffusion efficiency	---

LIST OF SUBSCRIPTS

Subscript	Definition
i	Inlet
p	Plug
min	Minimum
ext	External
eng max	At maximum engine area
o	Stagnation conditions
T	Total
x	Upstream of shock
y	Downstream of shock
1,2, etc.	Station locations

LIST OF SUPERSCRIPTS

Superscript	Definition
*	Indicates value at sonic conditions

DISCUSSION AND RESULTS

A. Extent of Supersonic Operation.

The primary requirement for a facility to have a supersonic flow capability is that a pressure ratio above 2.0 be available. The NATTS altitude test facility offers pressure ratios up to the order of 10.0. The only temperature requirement is that the stagnation temperature be great enough to prevent condensation of the air during expansion to the test section Mach number. For an air temperature of 180°F, condensation will occur at Mach numbers of 5.0 and 6.0. To prevent excessive moisture condensation during the supersonic expansion, the dew point of the air should be less than -10°F at atmospheric pressure (Reference 1). The facility's air conditioning process provides for water removal at air temperatures down to -30°F. With these conditions, a supersonic flow may readily be obtained. The size of the test section is a function of the facility airflow capacity and the test section Mach number.

A supersonic tunnel consists of a convergent-divergent nozzle which creates the supersonic flow; a constant area section which houses the test article; and, a diffuser which returns the flow to subsonic conditions. The tunnel may be of either rectangular (two dimensional) or circular (axially symmetric) geometry. The nozzle may be contoured to give a uniform parallel flow at its exit or it may have conical walls and offer some variation in Mach number. The diffuser may range from a constant-area (i.e., straight pipe) type to a variable-area device.

Operational considerations dictate the relation of the diffuser area to the nozzle area. Since the flow between the nozzle and diffuser is only approximately isentropic, the diffuser must have a slightly larger flow area than the nozzle. For stable diffuser operation, the throat must be further enlarged giving a slightly supersonic flow in the throat and a normal shock downstream of the throat. The process of starting the tunnel places a further restriction on the relation of diffuser to nozzle area. During the starting process, the stagnation pressure upstream of the nozzle is set to the desired value, and then the downstream portions of the tunnel are evacuated. As the pressure ratio across the nozzle increases beyond 2.0, a shock appears in the divergent section of the nozzle. Further increase in the pressure ratio causes the shock to move downstream till it is located in the test section. At this point the Mach number upstream of the shock is the maximum attainable in a given tunnel design and the stagnation pressure loss through the shock is the maximum

possible for the system. The area of the diffuser must be large enough to prevent the subsonic flow downstream of the shock from choking in the diffuser throat. Calculation of the downstream choking area is from:

$$A_x^* \cdot P_{o_x} = A_y^* \cdot P_{o_y}$$

Further increase in the pressure ratio moves the shock downstream to its design position. Since operation and starting of the tunnel require two different throat areas, either a fixed geometry diffuser sized to the starting requirement or a variable geometry diffuser must be used.

The maximum airflow that could be carried by a tunnel was a function of the exhaust facility capability when operating at the minimum pressure associated with the starting requirement, since the nozzle is choked during the starting and operating processes. The exhaust pressure was dependent on the test section Mach number. For a given test section Mach number, the minimum exhaust pressure during starting and the maximum airflow will be approximately the same for all diffuser configurations. As test section Mach number was varied for a given configuration, the stagnation pressure loss across the shock and the maximum available airflow also varied. This variation, for any configuration, was determined by considering the starting requirements and a straight pipe diffuser configuration. The results are shown in Figure 1. The calculations were made assuming an isentropic expansion followed by a normal shock and a subsonic diffusion. The adiabatic diffusion efficiency was 0.80. A stagnation temperature of 640°R was used with stagnation pressures of 55 and 35 in. HgA. Superimposed on Figure 1a are lines of constant test section diameter. Variation of stagnation temperature according to Mach number and standard altitude (Reference 2) for the maximum airflow and test section Mach number relation shown in Figure 1a resulted in a general increase in tunnel flow area, as shown in Figure 1b. Since the future 3E test cell in-line heater will make such temperatures available to that cell, its capacity was included in Figure 1b. Figure 2 shows the variation of airflow and test section diameter for tunnels which, in their original design, utilized 90 and 80 percent of the plant capacity. Both the cases of true temperature simulation and total temperature of 640°R are shown.

The purpose of the diffuser in a wind tunnel is to recover as much of the inlet stagnation pressure as possible. The load on the exhaust facility varied with the portion of the stagnation pressure which would be recovered. Three diffuser configurations

were studied:

Configuration (1)--Straight pipe.

Configuration (2)--Fixed geometry, convergent-divergent.

Configuration (3)--Variable geometry, convergent-divergent.

Configuration (1) was a straight pipe of area equal to the test section, with the shock located at the exit. The area of the fixed geometry, convergent-divergent diffuser was sized to 20 percent greater than A_y^* calculated for the starting requirement. The shock was considered as occurring at the Mach number associated with this area. The maximum area for the variable configuration was that corresponding to the starting requirement. Minimum area was sized so that the Mach number before the shock was 1.2. For each of the configurations, a subsonic divergent diffuser was located downstream of the shock.

A series of one dimensional analyses was made to determine the effect of diffuser configuration on the recovery of the inlet stagnation pressure for a range of test section Mach numbers. They were based on an assumed adiabatic flow with isentropic expansions and diffusion efficiencies varying with Mach number from Reference 3.

Initial Mach Number	Adiabatic Diffusion Efficiency
0.0 - 1.0	0.80
2.0	0.83
2.5	0.73
3.0	0.62
3.5	0.55
4.0	0.48
4.5	0.44
5.0	0.40

Appendix B shows the method used for calculation of stagnation pressure loss during diffusion.

The recoveries of configurations (2) and (3) were better than that of configuration (1). Consequently, their maximum available airflows were greater. Since tunnel airflow was determined by the starting condition, the net result of using configurations (2) and (3) was a reduction in percent of available airflow capacity being used. These results are shown in Figure 3.

The use of true temperature simulation in a tunnel sized for a given temperature--say 640°R--had a similar effect on the percent of available plant capacity used. As the temperature was increased, the airflow dropped but the pressure ratio across the tunnel remained constant so that there was no change in the maximum available airflow. The corresponding variation of temperature (i.e., test section Mach number) with percent of available plant capacity used is shown in Figure 3.

B. Research Conditions Available.

An indication of the flow regime present in a wind tunnel is the Reynolds number value per inch in the test section. Knowledge of this parameter indicates the places, if any, where laminar boundary layers might be present. The Reynolds number values per inch available are shown as a function of test section Mach number for a range of stagnation temperatures, at stagnation pressures of 55 and 35 in. HgA, in Figures 4a and 4b respectively. Assuming that laminar boundary layers may be found at Reynolds numbers, based on axial distance, of 750,000 or less, they will exist at Mach numbers above 3.0 and 1.75 for stagnation pressures of 55 and 35 in. HgA respectively. The in-line heater would be required to produce the stagnation temperatures needed. For a 640°R stagnation temperature, laminar boundary layers would exist only at a Mach number of 5.0 and a 35 in. HgA stagnation pressure.

C. Design Point for Research Tunnel.

Due to its simplicity, a tunnel with a straight pipe diffuser was chosen for initial studies and consideration. The design point Mach number and airflow were 3.0 and 300 pounds per second, respectively, with a stagnation pressure of 55 in. HgA. The variation of several pertinent parameters with axial tunnel location is shown in Figures 5a and 5b for a stagnation temperature of 640°R and for true temperature simulation respectively. The values of Mach number of 3.0 and airflow of 300 pounds per second were chosen for demonstration because they offered a useful supersonic Mach number as well as taking full advantage of the facility capacity. Another attractive design from a research point of view was a Mach number of 5.0 with a much lower airflow.

D. Simulation of Flight Conditions.

Use of a constant inlet stagnation pressure restricts testing to a fixed Mach number versus altitude variation as shown in Figure 6.

The altitudes are those associated with the free stream static pressure at the given Mach number. They may be affected to some extent by the presence of the engine in the tunnel. Variation of inlet stagnation pressure from 55 to 35 in. HgA increased altitude by approximately 10,000 feet.

Only a very small portion of Figure 6 is enclosed by a stagnation temperature of 640°R. Using the 3E test cell in-line heater, true simulation may be obtained at Mach numbers up to 3.55.

E. Accommodation of Air Breathing Engines.

The previously mentioned one-dimensional analysis was extended to consider the size air breathing engine which might be accommodated in a supersonic tunnel. Both ramjet and turbojet engines were considered, with the primary difference between the two being the method by which airflow was estimated. Inlet area and free stream stagnation conditions were used for ramjet airflow, while sea level static airflow was corrected to free stream stagnation conditions for turbojet airflow.

The primary effects of the presence of an engine were the blockage effects due to the reduced flow area around the engine. The reduction in area must not be sufficient to choke the flow around the engine. As before, the starting process determined the minimum flow area downstream of the nozzle exit. For a tunnel with a straight pipe diffuser, the minimum flow area occurred at the maximum engine diameter. Using the assumptions previously stated, an equation was derived for the variation of maximum engine diameter with test section Mach number (see Appendix C). The following assumptions were made considering ramjet engine geometry:

(1) Ratio of inlet area (i.e., capture area) to maximum engine area (α) was equal to 0.90.

(2) Ratio of inlet plug area to maximum engine area (β) was equal to 0.81.

The maximum engine area based on the starting requirements with the exhaust plant operating at 90 percent of capacity was calculated for two limiting conditions:

(1) The engine inlet was choked or was passing its maximum airflow for a given set of stagnation conditions. (Since the

exhaust nozzle would be sized to choke during engine operation, the engine inlet may well choke when passing cold flow.)

(2) There was zero engine airflow or $\alpha = \beta$. The results of these calculations for stagnation pressure and temperature of 55 in. HgA and 640°R are shown in Figure 7a, by the curve with test-section-to-nozzle-exit-area ratio of 1.0. The maximum engine area for this case ranged up to slightly less than five feet square at a test section Mach number of 4.0 to 5.0.

The type of free jet installation which has a test section to nozzle exit area ratio of 1.0 is known as a closed jet. In order to obtain larger maximum engine areas, test section to nozzle exit area ratio was increased to 1.2 and then to 1.4 and new calculations made. This type of free jet installation is denoted as an open jet. The results of these calculations are also shown in Figure 7a. The open jet installation permitted considerably larger maximum engine areas. The span between the results of the two limiting calculations increased as the test section to nozzle exit area ratio was increased. For the zero engine airflow limit, the increased maximum engine area reflects the increase in test section while the flow area remained constant. The increased maximum engine area for the choked engine inlet condition was due to the increased engine airflow and reduced external flow as test section area was increased. Figure 7b shows similar results but using true temperature simulation.

The size of turbojet engine that may be accommodated was estimated by studying the requirements of the J52 and J79 engines. The airflow was calculated at the desired flight conditions. The inlet area needed to pass these flows was then computed from the airflow and the free stream conditions. Inlet area was assumed to be 90 percent of the maximum engine area. Maximum engine area as a function of test section Mach number for the J79 and J52 engines is plotted in Figure 7b. Due to the physical size of the engine, the maximum engine area was given a lower limit of 7.5 square feet. These calculations were restricted to true temperature conditions.

An inherent disadvantage in the open jet-straight pipe diffuser configuration was the large increase in flow area downstream of the engine. This resulted in a higher Mach number upstream of the shock which increased the stagnation pressure loss through the shock. This effect was revealed during studies made for an engine tunnel sized to a 300-pound-per-second flow, at Mach numbers of 2.0, 2.5, and 3.0. At a test section Mach number of 3.0, the Mach number downstream of the engine increased to 3.32, and the

associated capacity of the exhaust facility was below 300 pounds per second. No problem existed for Mach numbers of 2.0 and 2.5. For a given airflow, the use of an open jet configuration to allow a larger size test article was associated with a reduction in Mach number capability. The results of these studies are shown in Figures 8a to 8c.

F. Design Point Engine Tunnel.

The maximum engine area restrictions corresponding to a closed jet configuration require the use of an open jet configuration for all but the smallest of engines. Since airflow capacity is of primary importance in engine testing, it was decided to consider a Mach number 2.5, 300-pound-per-second tunnel as a design point tunnel. Figure 8b gives variation of a few aerodynamic parameters through the tunnel. True temperature simulation would be required to obtain useful engine data, and a stagnation pressure of 55 in. HgA is necessary to obtain sufficient airflow.

The method of area variation between the nozzle exit and the test section in an open jet configuration is a problem in two-dimensional or axially-symmetric flow, and as such, may be studied with the method of characteristics. In order to eliminate losses and to maintain a sufficiently low static pressure at the engine exhaust, a criterion that the flow around the engine remain supersonic was established. This criterion placed certain restrictions on the initial turning angle of the flow as it passed the engine. If this angle was too large, the flow could not sustain a similar turn back to its original direction without becoming subsonic. The magnitude of the turning angle depended on the wedge angle of the engine cowl lip from the free stream flow angle and on the tunnel wall angles. One sample two-dimensional characteristics study was made for a free stream Mach number of 3.0. In order to satisfy the criterion mentioned, the engine cowl lip angle was limited to approximately twenty degrees, while the initial tunnel wall angle was ten degrees, followed by two five-degree turns back to the original direction. This was only one of many methods that could be used to vary the area and still obey the criterion stated. A method satisfactory at Mach number of 3.0 would also be satisfactory at Mach number of 2.5, although less restrictive methods might also work.

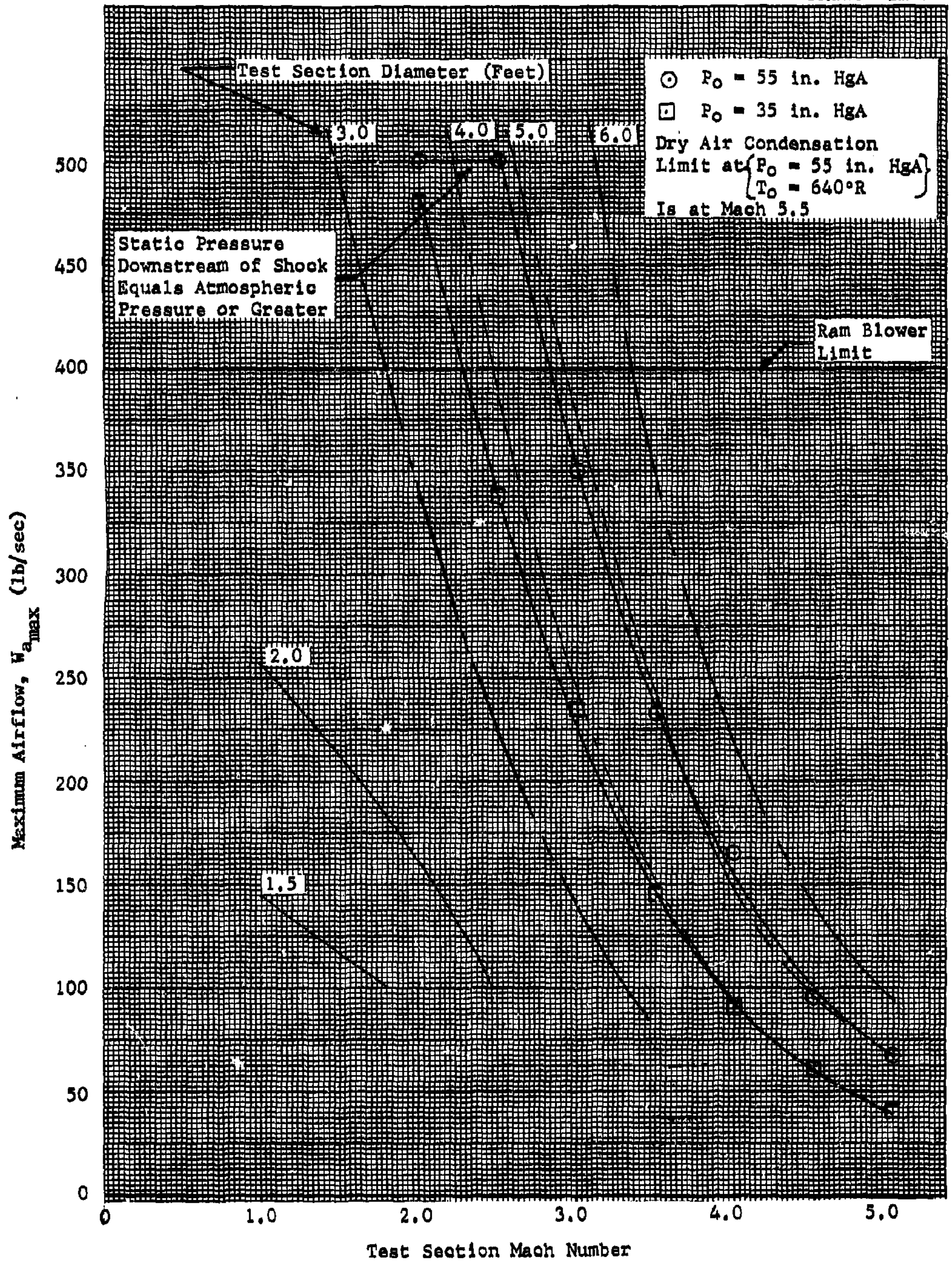

Figure 1a: Extent of Supersonic Operation Available with the Present NATTS Facilities at a Stagnation Temperature of 640°R

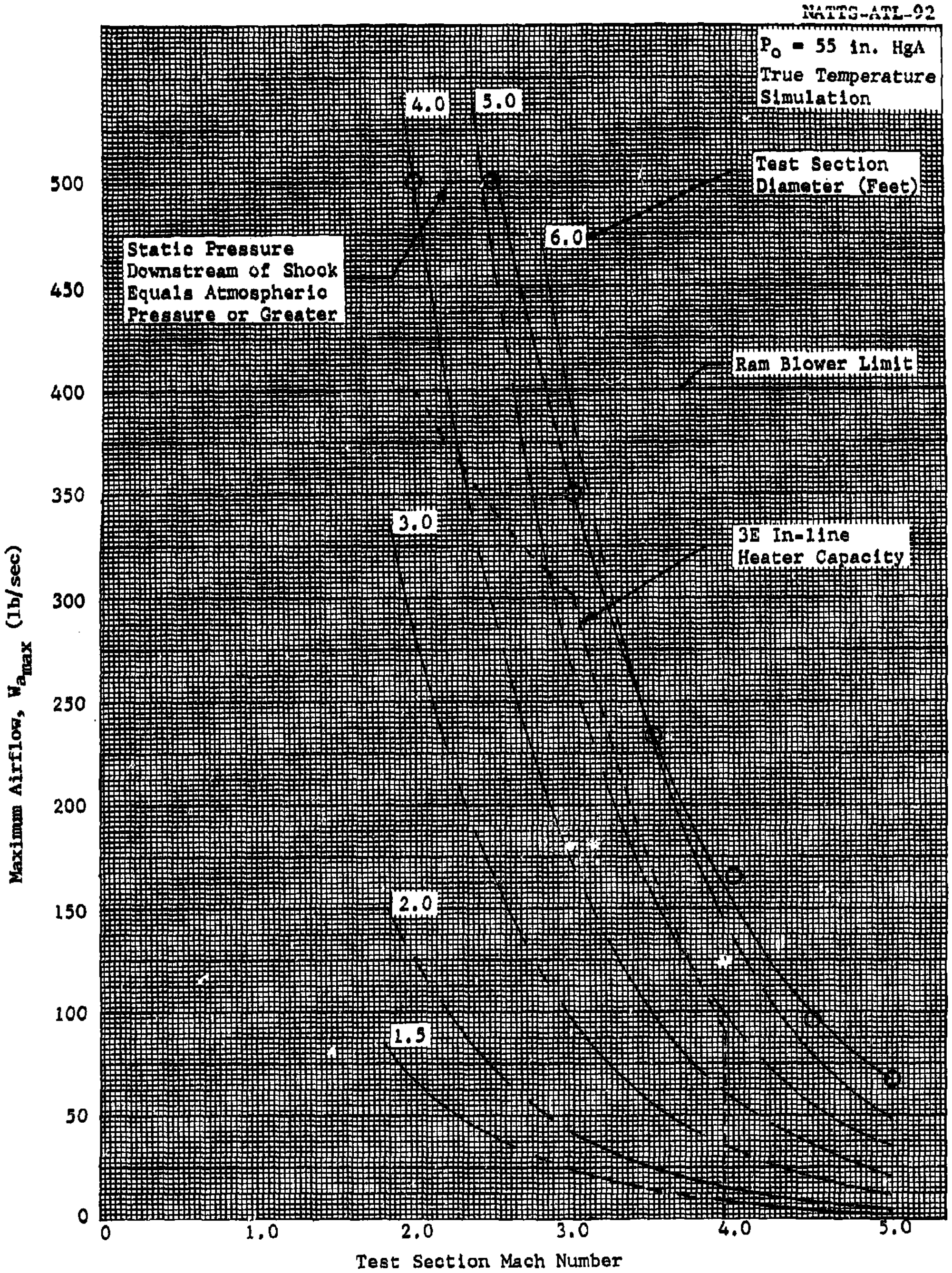

Figure 1b: Extent of Supersonic Operations Available Using the Present NATTS Facilities with True Temperature Simulation

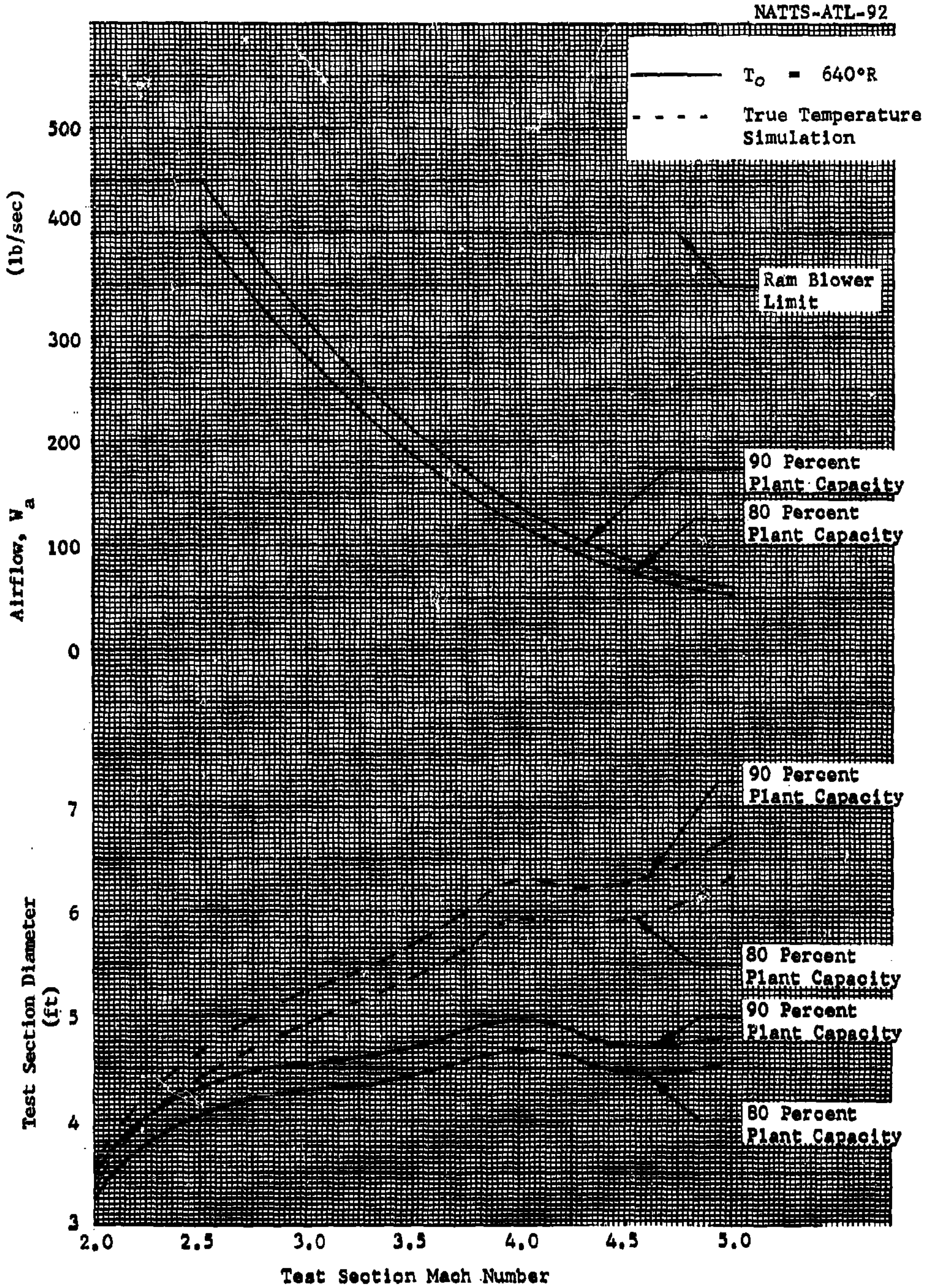

Figure 2: Effect of Mach Number on Airflow and Test Section Diameter for Various Plant Operating Levels

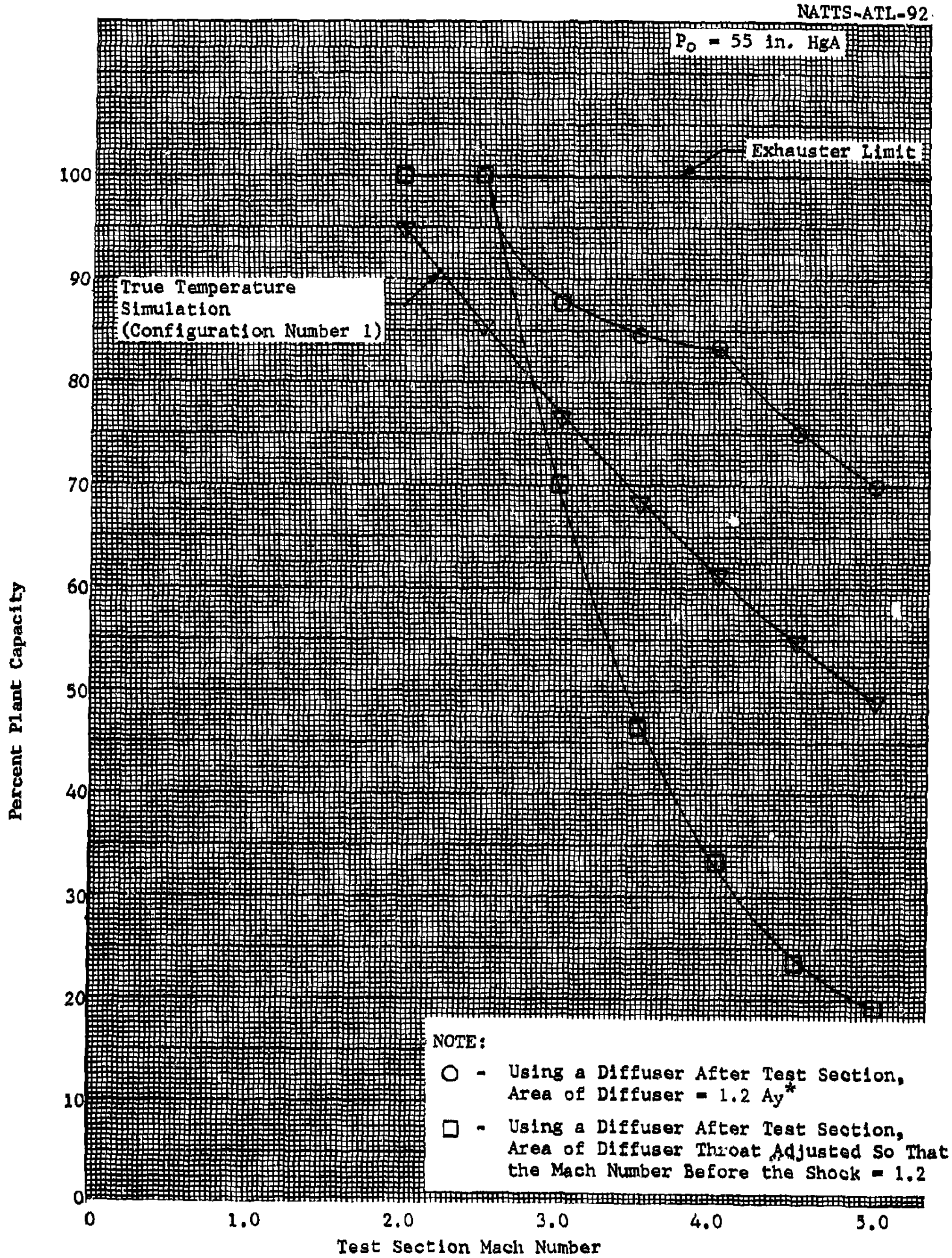

Figure 3: Effect of Various Diffuser Configurations and Use of True Temperature Simulation on Percent of Plant Capacity Required

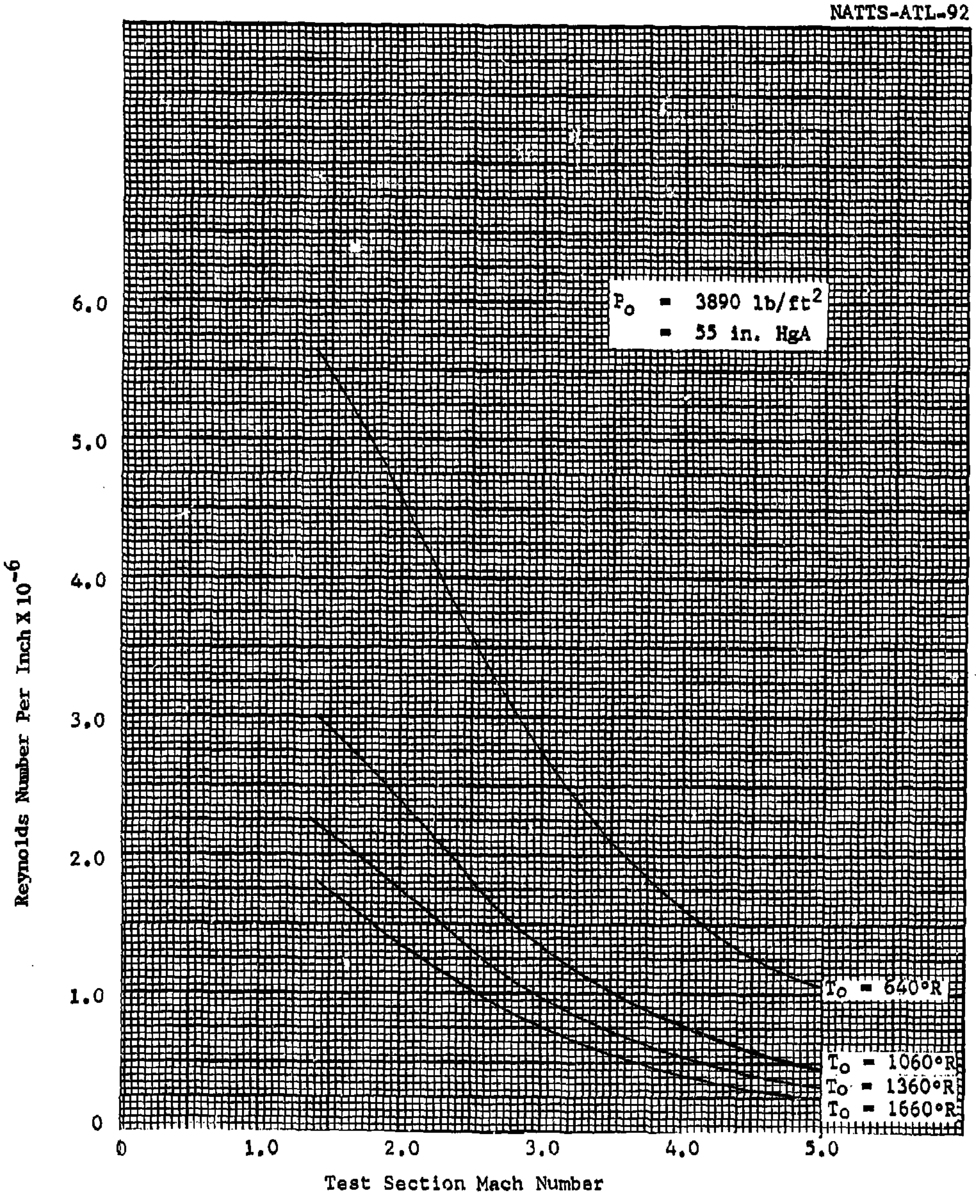

Figure 4a: Effect of Stagnation Temperature on Reynolds Number Per Inch for a Range of Test Section Mach Numbers at a Stagnation Pressure of 55 in. HgA

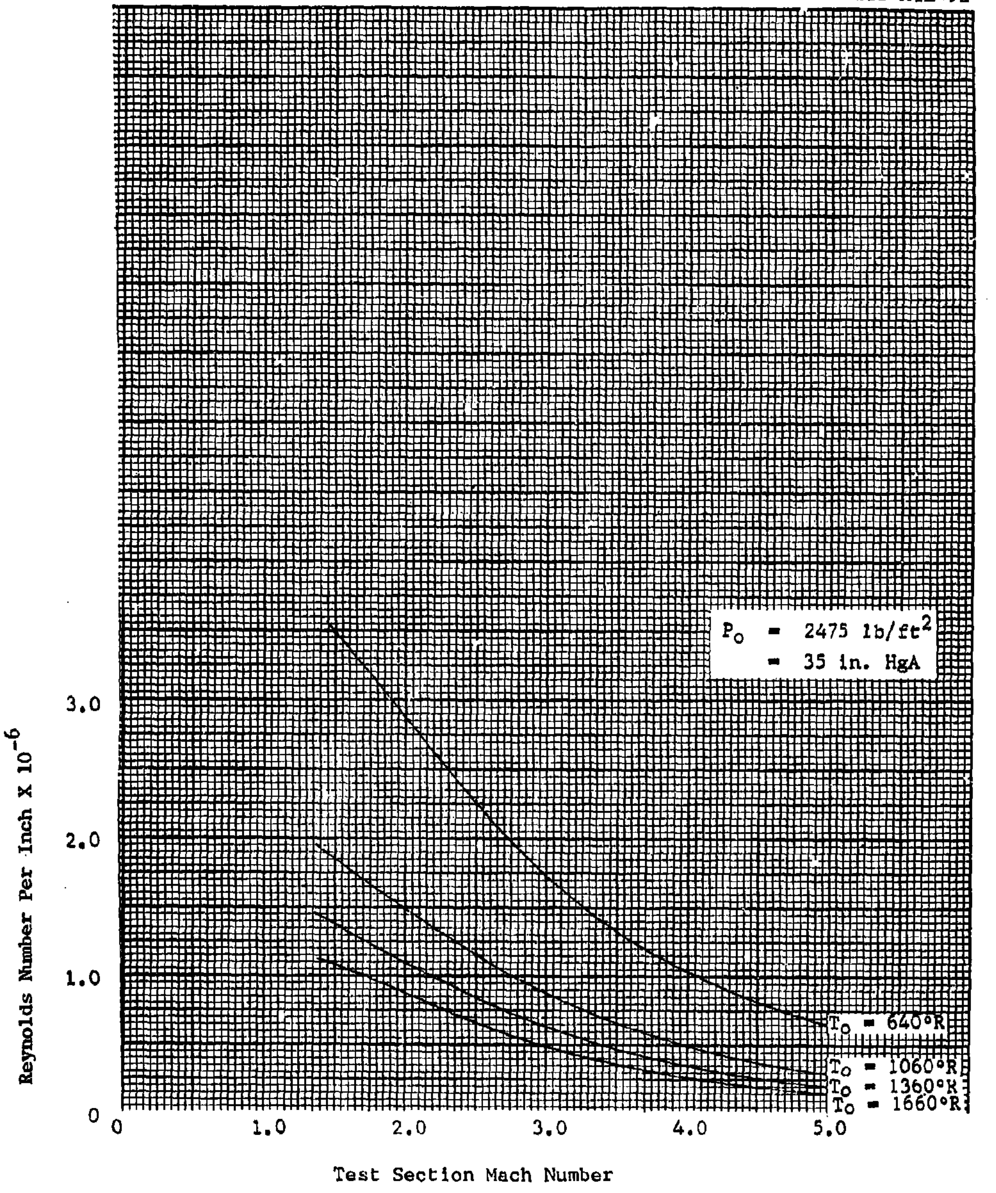

Figure 4b: Effect of Stagnation Temperature on Reynolds Number Per Inch for a Range of Test Section Mach Numbers at a Stagnation Pressure of 35 in. HgA

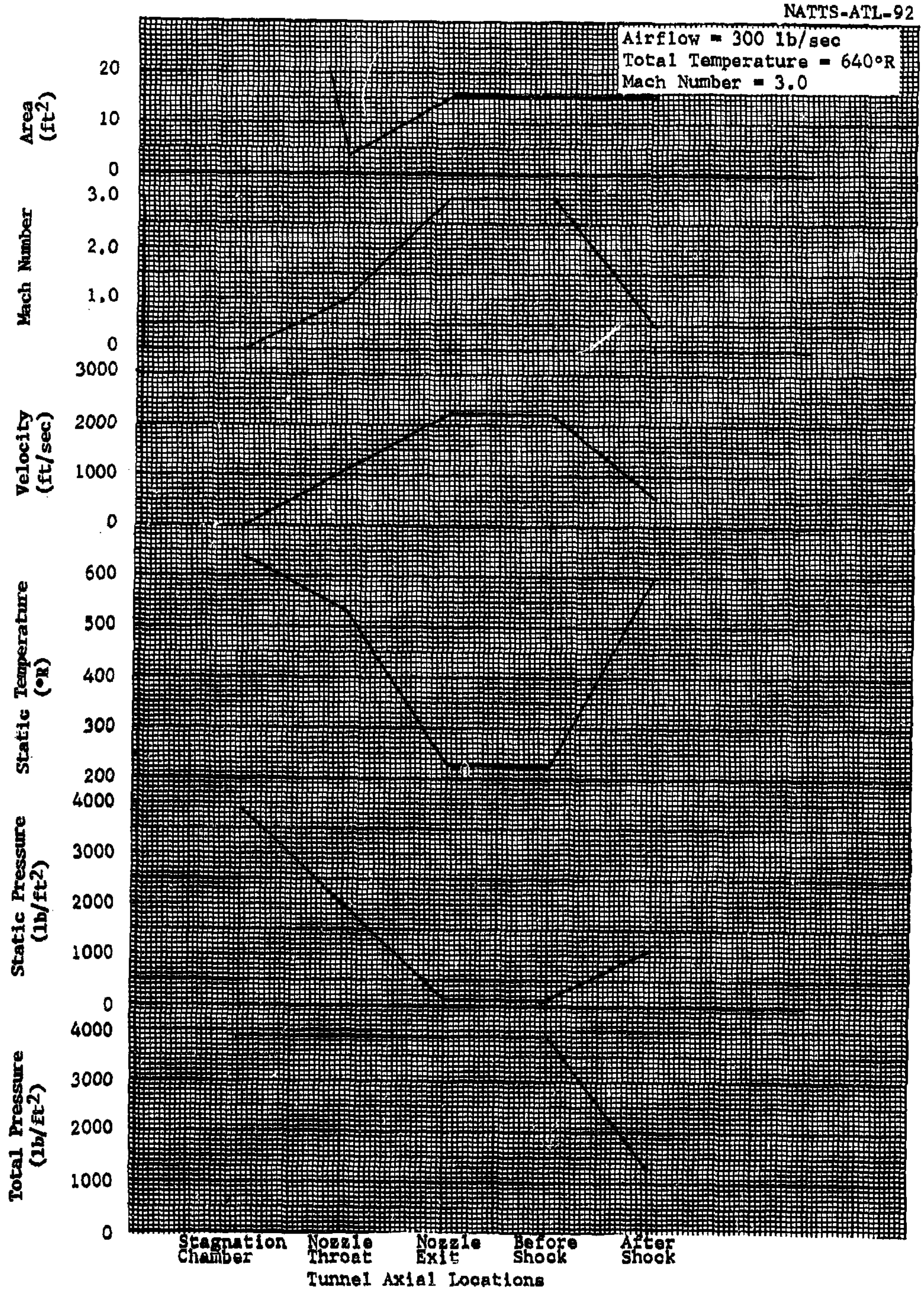

Figure 5a: Variation of Parameters in a Design Point Research Tunnel Using True Temperature of 640°R

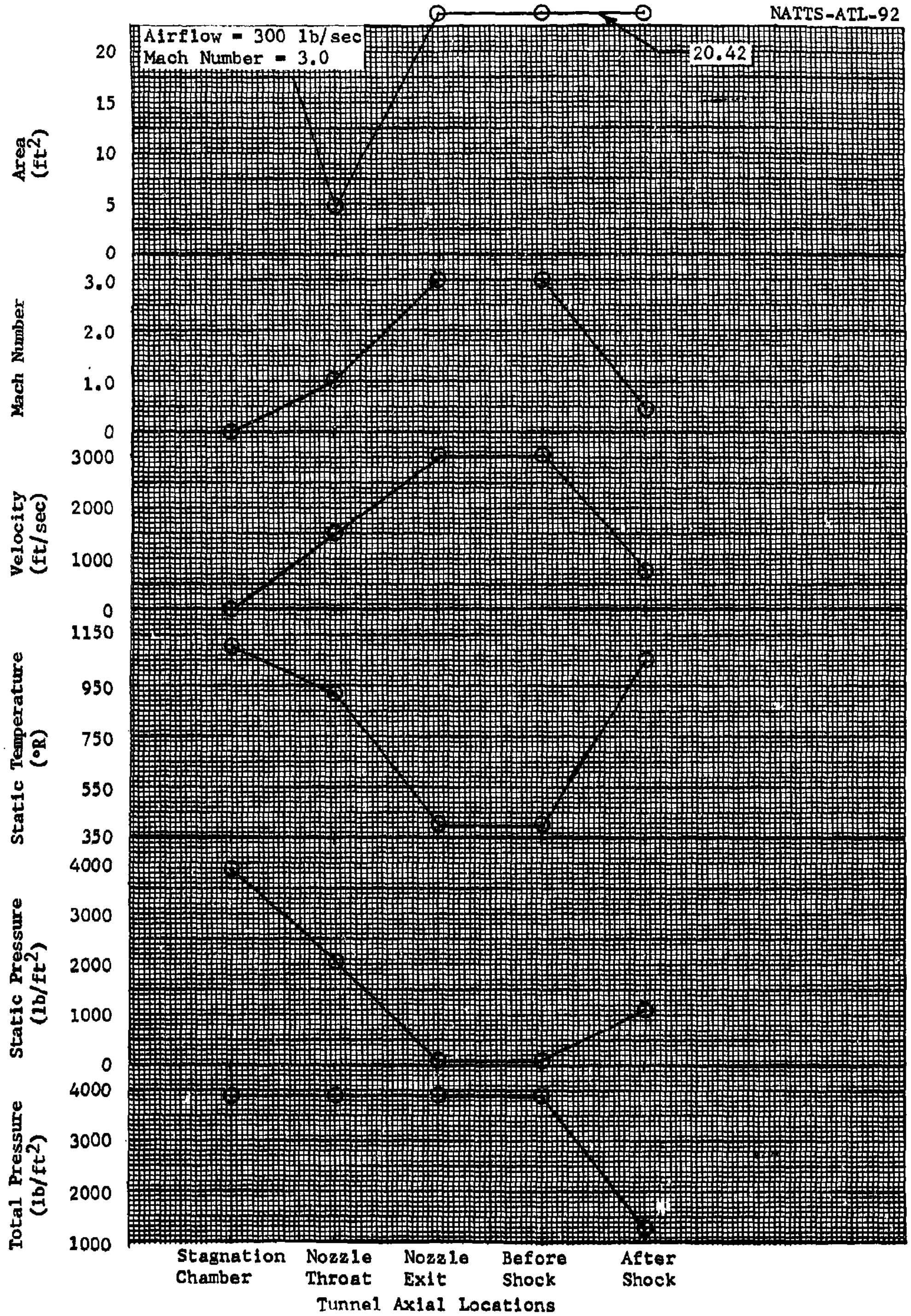

Figure 5b: Variation of Parameters in a Design Point Research Tunnel Using True Temperature Simulation

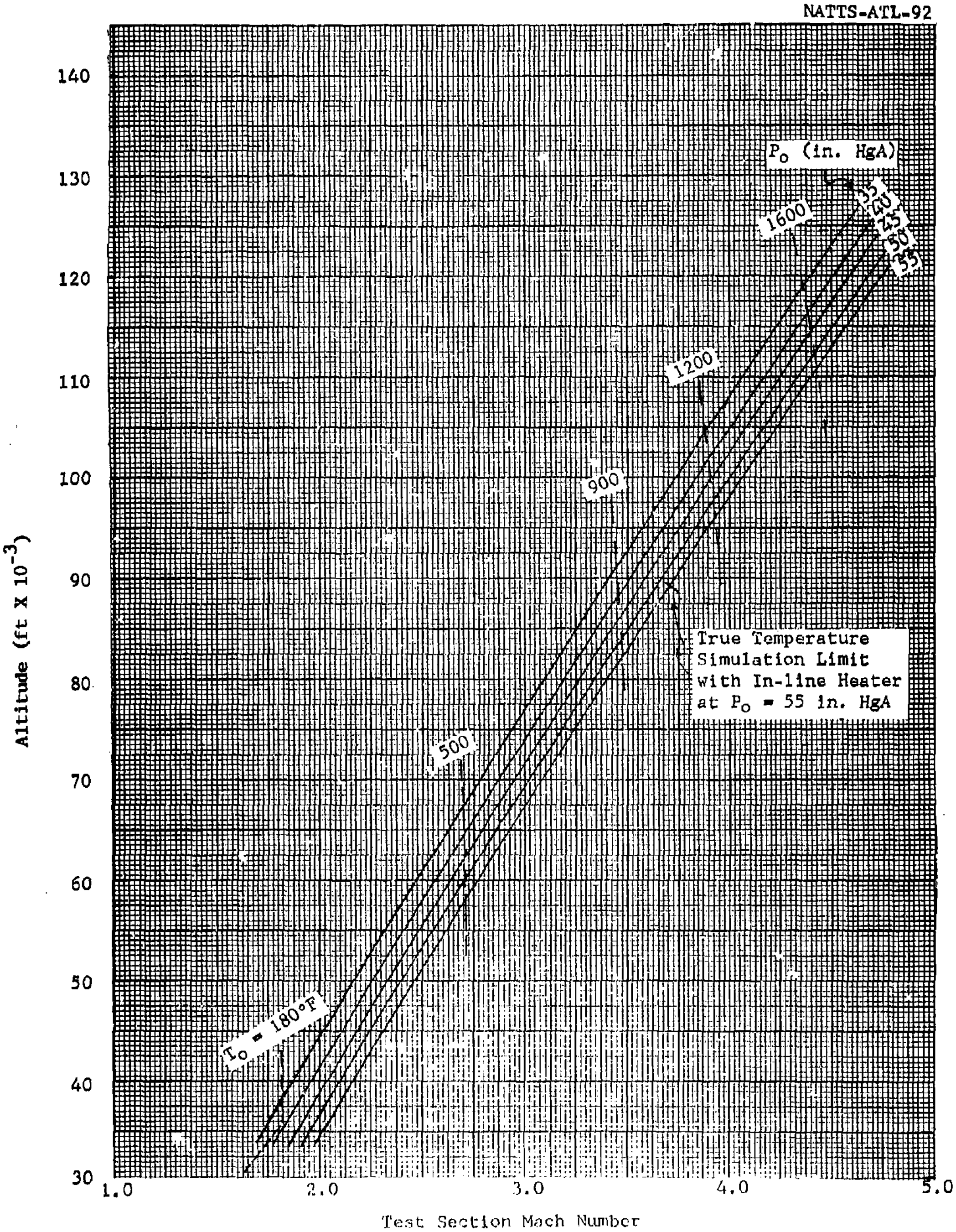

Figure 6: Flight Conditions Available with a Given Inlet Stagnation Pressure

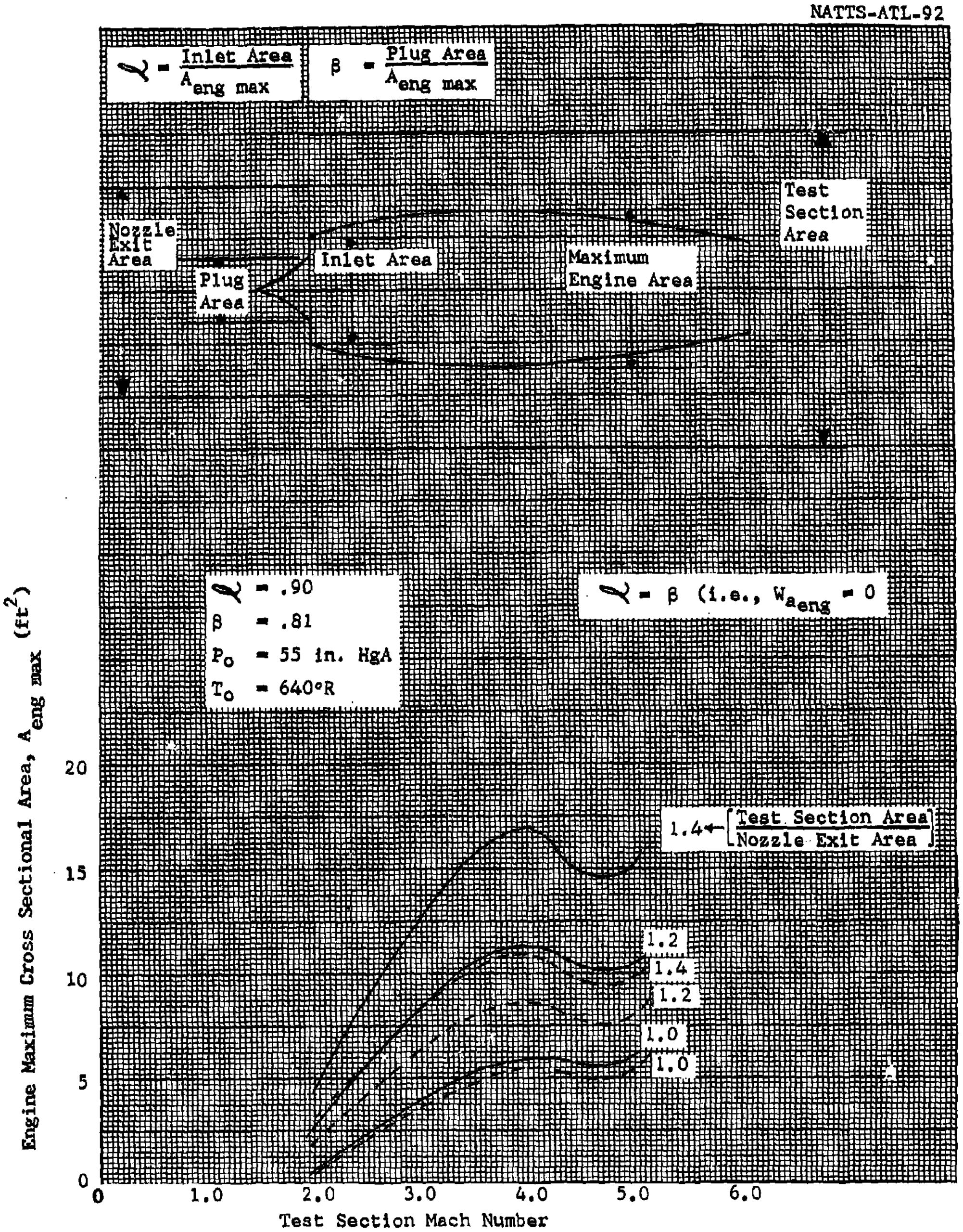

Figure 7a: Maximum Engine Cross Sectional Areas for Various Test Section Mach Numbers and Configurations at a Stagnation Temperature of 640°R

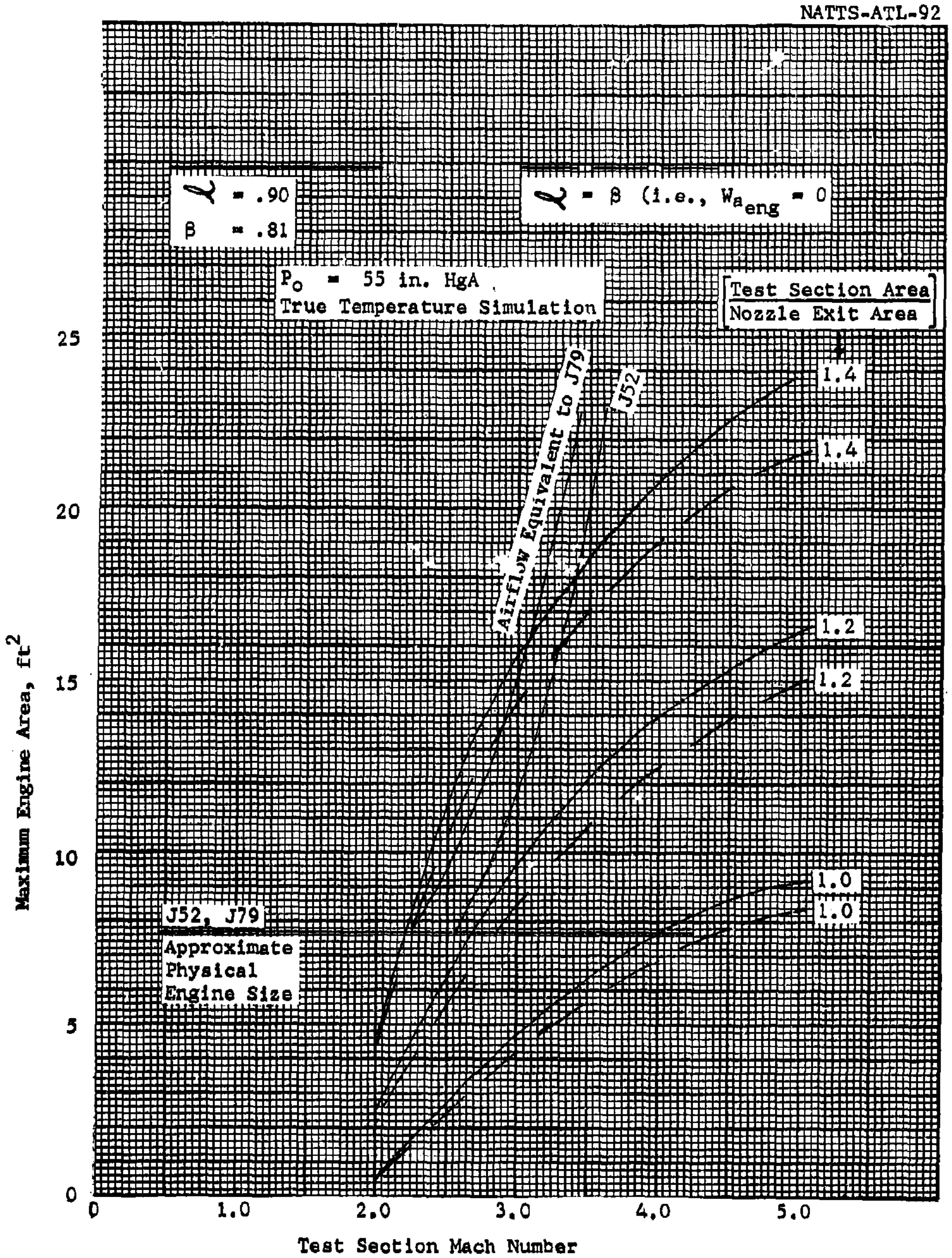

Figure 7b: Maximum Engine Cross Sectional Areas for Various Test Section Mach Numbers and Configurations Using True Temperature Simulation

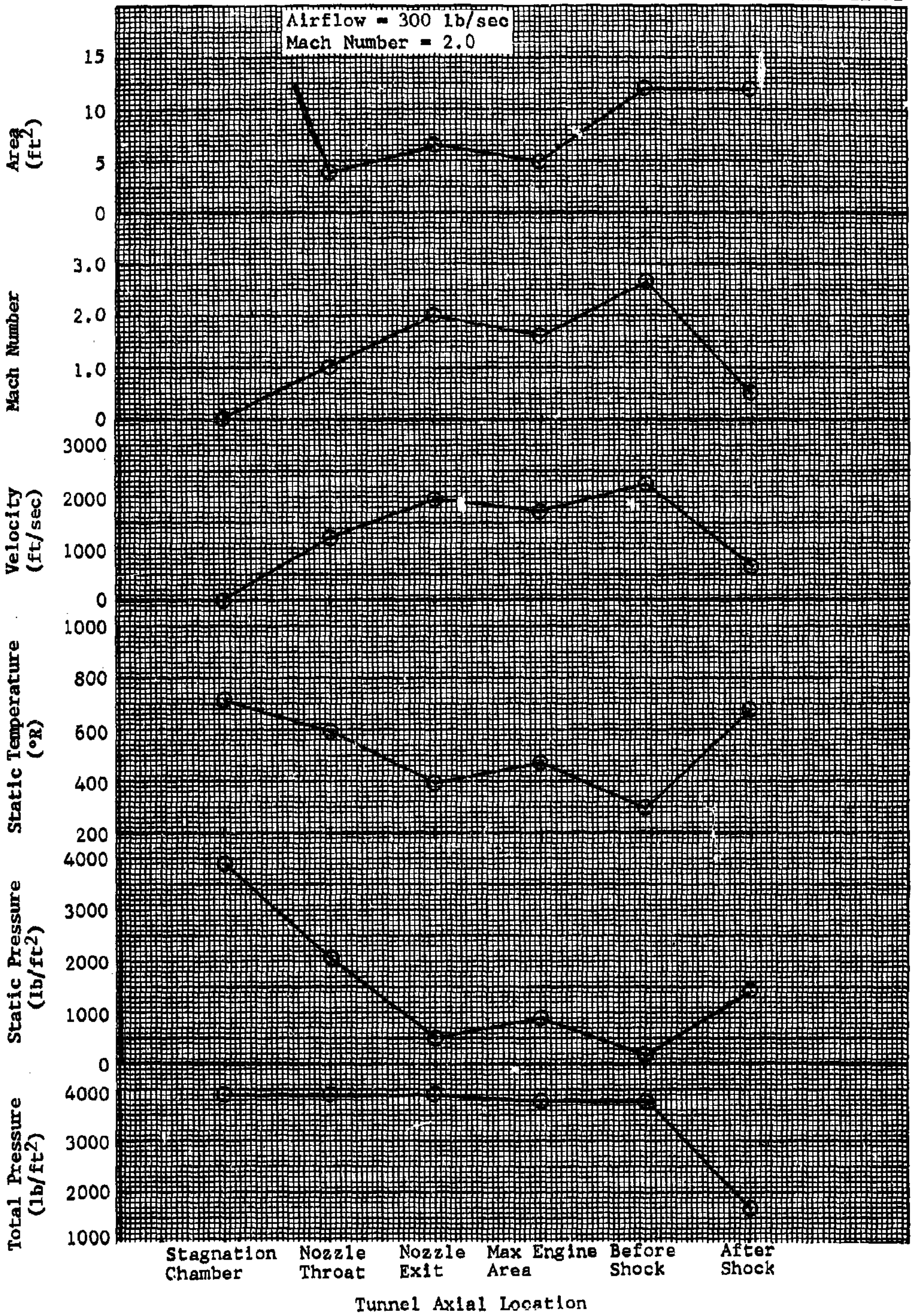

Figure 8a: Parameter Variation within a Mach Number 2.0 Design Point Engine Tunnel Using True Temperature Simulation

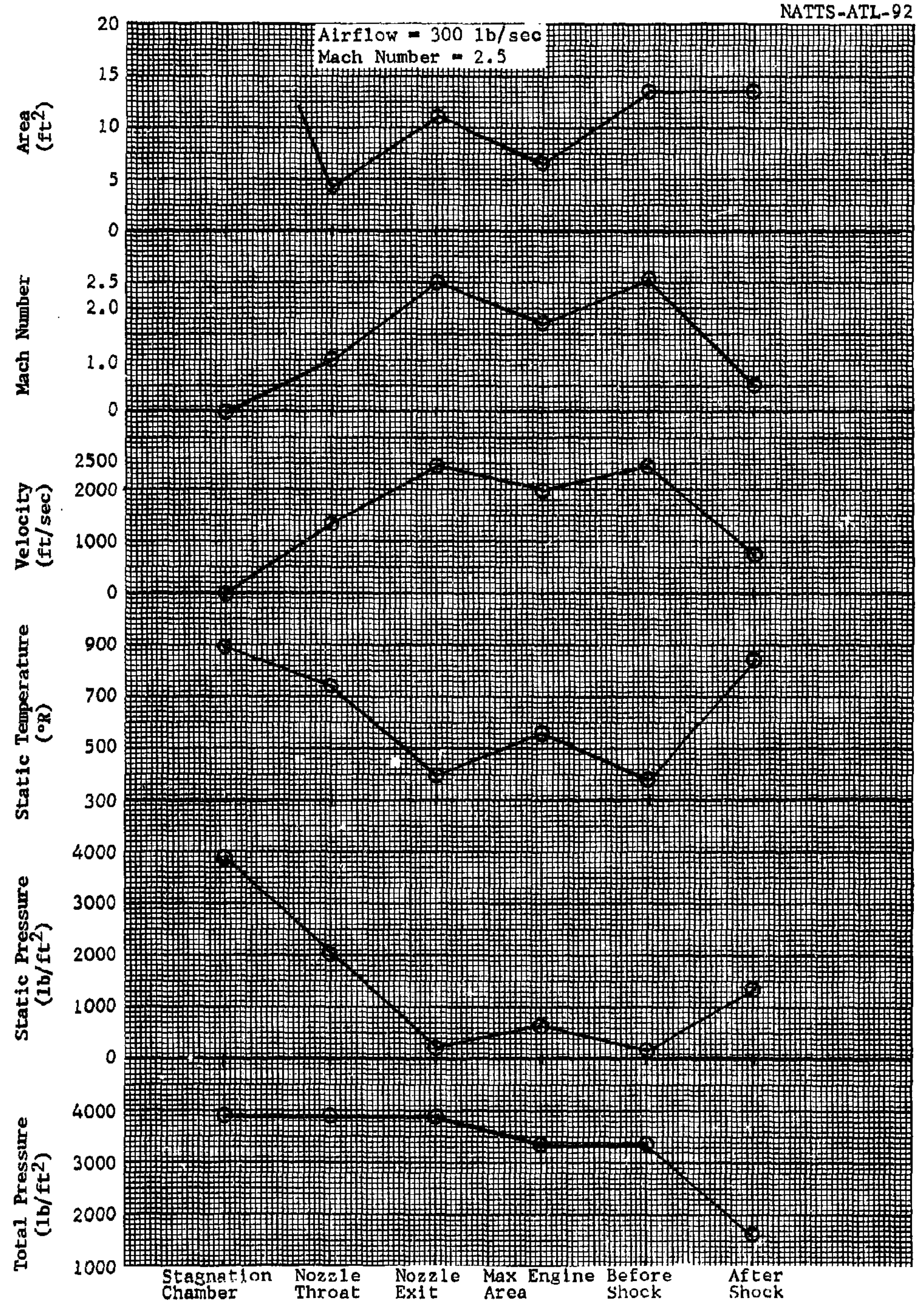

Figure 8b: Parameter Variation within a Mach Number 2.5 Design Point Engine Tunnel Using True Temperature Simulation

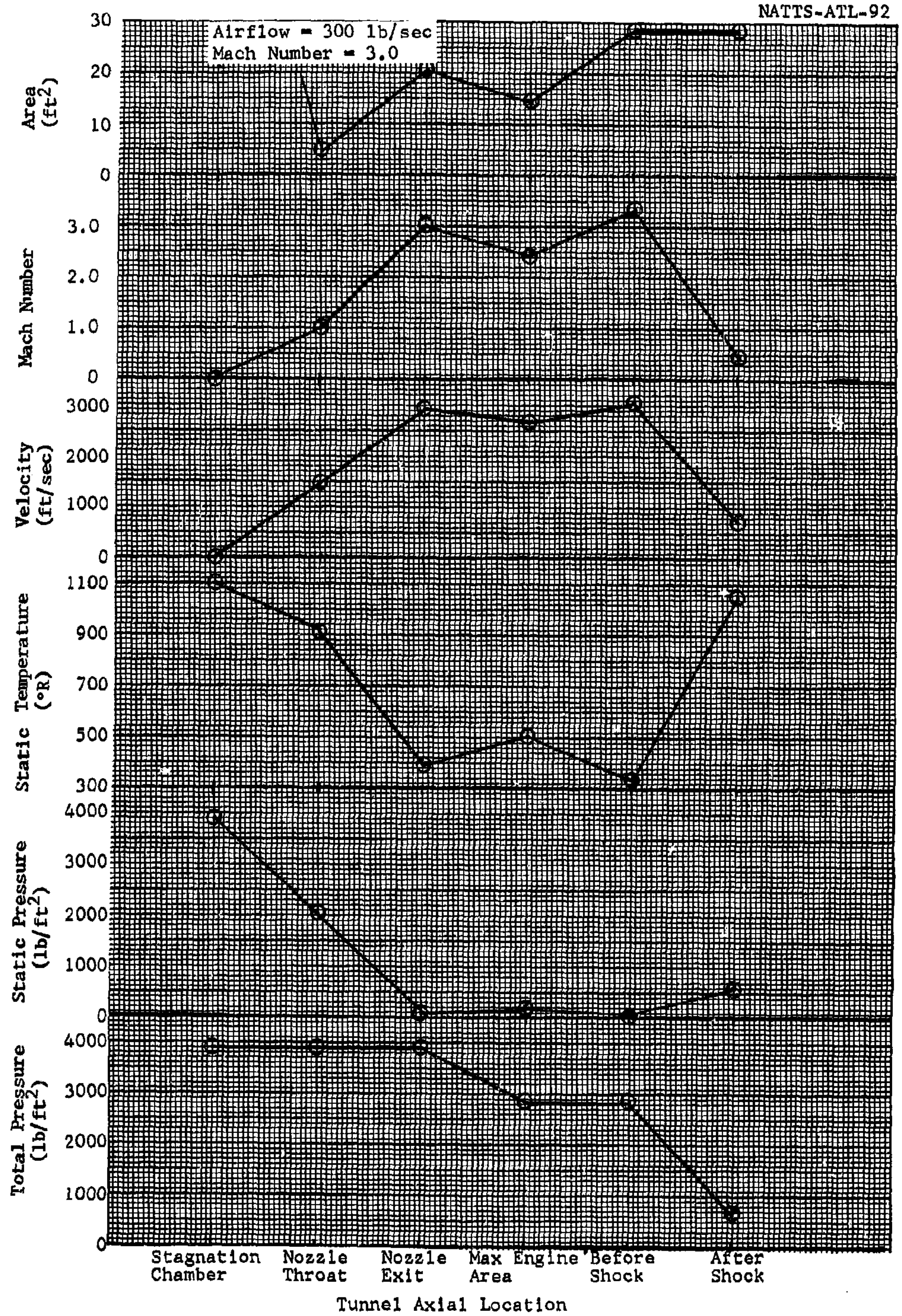

Figure 8c: Parameter Variation within a Mach Number 3.0 Design Point Engine Tunnel Using True Temperature Simulation

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2. Book - U. S. Standard Atmosphere, 1962. (National Aeronautics and Space Administration, United States Air Force, and United States Weather Bureau, December 1962)

3. Book - Shapiro, Ascher H. Dynamics and Thermodynamics of Compressible Fluid Flow. (The Ronald Press Company, 1953)

APPENDIX A

FOUNDATIONAL RESEARCH PROJECT PROPOSAL--FISCAL YEAR 1964

1. TITLE OF PROJECT:

Investigation of the Research Capabilities of the NATTS Altitude Test Facilities

Part 1: Investigate and define the research flow capabilities of the NATTS altitude test facilities

Part 2: Design and construction of a two-dimensional supersonic wind tunnel

2. COGNIZANT DEPARTMENT:

Aeronautical Turbine Laboratory

3. BUDGET ESTIMATE:

	Labor	Material/ Services
a. Design a two-dimensional supersonic nozzle, test section, and diffuser considering aerodynamic and mechanical problems (including drafting)............	$1000	
b. Construct items under paragraph a. above....................................	3000	$1000
c. Design, construct, and install necessary instrumentation for calibration of tunnel.................................	1000	300
d. Install in altitude cell.................	1000	200
e. Continuous engineering services..........	1300	
	$7300	$1500

TOTAL BUDGET ESTIMATE: $8800.

4. RESEARCH PLAN:

a. Specific aims:

(1) Continue the investigation and definition of the research capabilities of the NATTS altitude test facilities.

(2) Design, construction, and calibration of a two-dimensional supersonic wind tunnel.

b. Methods of procedure:

(1) Continue survey of available literature to ascertain airflow requirements for free-jet testing of various available air-breathing powerplants.

(2) Develop aerodynamic designs for two-dimensional supersonic wind tunnel.

(3) Mechanical design, construction, and calibration of the supersonic wind tunnel.

5. SUPPORTING DATA:

a. Significance of research:

The definition of a supersonic flow capability over a well defined range is prerequisite for establishing NATTS research abilities in the field of aerodynamics and propulsion, utilizing existing facilities. Such a capability would be invaluable in the following areas:

(1) Research and development of supersonic air-breathing powerplants.

(2) Aerodynamic research in the transonic and supersonic regimes.

6. INVESTIGATORS:

a. Principal Investigator--W. Richard Vollmar (GS-861-12), Weptask Division "C", Aeronautical Turbine Laboratory.

b. Consultant--Martin E. Hoyer (GS-861-14), Chief of Weptask Division "C", Aeronautical Turbine Laboratory.

7. PREVIOUS WORK DONE ON THIS PROGRAM:

a. In Fiscal Year 1963 the supersonic capabilities of the present NATTS altitude test facilities were defined over a range of test section Mach numbers for various plant capability limits. Results indicated that the most favorable configuration for this tunnel would be:

(1) Airflow: 250 pounds per second.

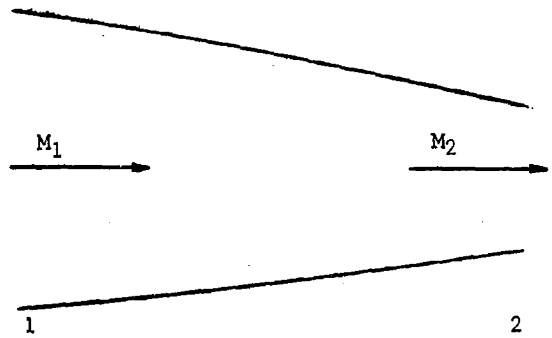
(2) Test section Mach number: 3.0.

b. No organization other than NATTS has been approached for funding of this project.

c. This project is a continuation of previously approved Foundational Research Project Number 11.

APPENDIX B

CALCULATION OF STAGNATION PRESSURE LOSS
THROUGH A SUPERSONIC ADIABATIC DIFFUSION
FOR A KNOWN ADIABATIC DIFFUSION EFFICIENCY



The following quantities are known at station 1:

$$A_1, A_1^*, M_1, T_{o_1}, T_1, P_{o_1}, p_1$$

At station 2 only the area (A_2) is known.

Since flow is adiabatic, $T_{o_1} = T_{o_2}$.

The Mach number for an isentropic diffusion between station 1 and station 2 may be calculated by:

$$M_{2\ isen} = f\ (A_2/A_1^*)$$

$$A_1^* = A_2^*{}_{isen}$$

The static temperature after the isentropic expansion may be obtained from:

$$T_{2\ isen} = f\ (M_{2\ isen}, T_{o_1})$$

The actual static temperature is then:

$$T_2 = \frac{T_{2\ isen} - T_1}{\eta} + T_1$$

The actual Mach number may then be calculated by:

$$M_2 = f\ (T_2,\ T_{o_1})$$

The actual choking area A_2^* is then:

$$A_2^* = f\ (M_2,\ A_2)$$

For any flow:

$$\frac{W_a \sqrt{T_o}}{A^* P_o} = K \qquad \text{or} \qquad \frac{W_{a_1} \sqrt{T_{o_1}}}{A_1^* P_{o_1}} = \frac{W_{a_2} \sqrt{T_{o_2}}}{A_2^* P_{o_2}}$$

For this case,

$$W_{a_1} = W_{a_2}$$

$$T_{o_1} = T_{o_2}$$

$$A_1^* P_{o_1} = A_2^* P_{o_2}$$

The stagnation pressure at station 2 is then:

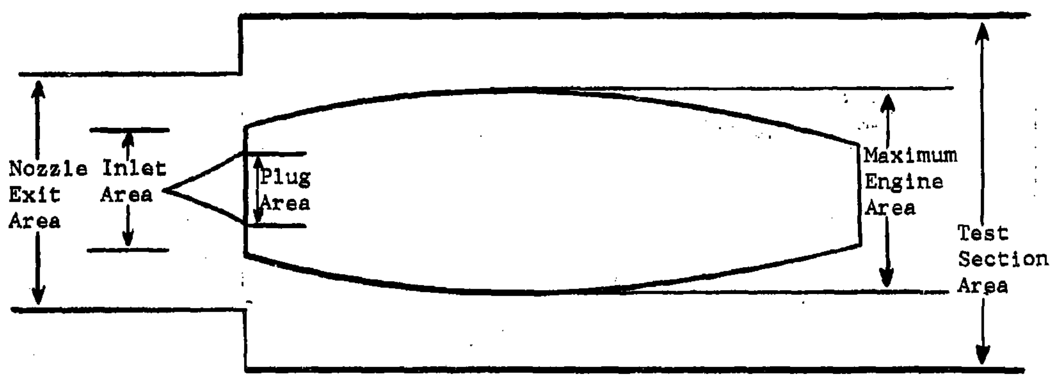
$$P_{o_2} = f\ (P_{o_1},\ A_1^*,\ A_2^*)$$

and the static pressure is:

$$P_2 = f\ (M_2,\ P_{o_2})$$

APPENDIX C

DERIVATION OF AN EQUATION FOR MAXIMUM ENGINE CROSS SECTIONAL AREA AS A FUNCTION OF NOZZLE EXIT MACH NUMBER AND TUNNEL STARTING REQUIREMENTS



$$\alpha = \frac{\text{Inlet Area}}{A_{eng\ max}} = \frac{A_i}{A_{eng\ max}}$$

$$\beta = \frac{\text{Plug Area}}{A_{eng\ max}} = \frac{A_p}{A_{eng\ max}}$$

$$W_{a_T} = W_{a_{ext}} + W_{a_{eng}}$$

$$W_{a_{ext}} = W_{a_T} - W_{a_{eng}}$$

If engine inlet chokes:

$$W_{a_{eng}} = K \frac{P_{o_y}}{\sqrt{T_{o_y}}} A_{min}$$

where:

$$A_{min} = \gamma (A_i - A_p)$$

where:

$$\gamma = \frac{\text{Minimum engine inlet flow area}}{A_i - A_p}$$

$$A_{min} = \gamma A_i (1 - A_p/A_T)$$

$$W_{a_{eng}} = K \frac{P_{o_y}}{\sqrt{T_{o_y}}} \gamma A_i (1 - A_p/A_i)$$

External airflow may be calculated from:

$$\frac{W_{a_{ext}} \sqrt{T_{o_y}}}{(A_T - A_{eng\ max})^* P_{o_y}} = K$$

where:

$$(A_T - A_{eng\ max})^* = \frac{W_{a_{ext}} \sqrt{T_{o_y}}}{K P_{o_y}}$$

To prevent choking of flow at maximum engine area:

$$(A_T - A_{eng\ max}) = 1.15 \frac{W_{a_{ext}} \sqrt{T_{o_y}}}{K P_{o_y}}$$

$$\therefore W_{a_{ext}} = \frac{K P_{o_y}}{1.15 \sqrt{T_{o_y}}} (A_T - A_{eng\ max})$$

Now:

$$W_{a_T} = \frac{K P_{o_y}}{\sqrt{T_{o_y}}} \left[\frac{A_T - A_{eng\ max}}{1.15} + \gamma \alpha A_{eng\ max} (1 - \beta/\alpha) \right]$$

Rearranging:

$$\frac{W_{a_T}\sqrt{T_{o_y}}}{K\,P_{o_y}\,A_T} = \frac{1 - (A_{eng\ max}/A_T)}{1.15} + \eta\,(\alpha - \beta)\,\frac{A_{eng\ max}}{A_T}$$

$$\frac{A_{eng\ max}}{A_T} = \left[\frac{1}{1.15\,\eta\,(\alpha - \beta) - 1}\right]\left[\frac{1.15\,W_{a_T}\sqrt{T_{o_y}}}{K\,P_{o_y}\,A_T} - 1\right]$$

For zero engine airflow, $\alpha = \beta$

$$\frac{A_{eng\ max}}{A_T} = -\left[\frac{1.15\,W_{a_T}\sqrt{T_{o_y}}}{K\,P_{o_y}\,A_T} - 1\right]$$

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